Copy
RM L53I22b

NACA

RESEARCH MEMORANDUM

DRAG DUE TO LIFT AT MACH NUMBERS UP TO 2.0

By Edward C. Polhamus

Langley Aeronautical Laboratory
Langley Field, Va.

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON
November 18, 1953

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

DRAG DUE TO LIFT AT MACH NUMBERS UP TO 2.0

By Edward C. Polhamus

INTRODUCTION

In references 1 and 2 it has been shown that, if the "area rule" is utilized properly, it is possible to obtain values of zero-lift drag which, for a wide variety of wing-fuselage configurations, approach that for the basic fuselage alone. This fact makes the selection of a wing less dependent on its zero-lift drag and therefore allows a wider range of wings to be considered with regard to drag due to lift. The purpose of this paper therefore is to discuss the effect of wing geometry on the drag due to lift at Mach numbers up to 2.0 and the effect of application of the area rule on the drag at lifting conditions.

In figure 1 a typical variation of the drag with lift coefficient for a plane, or flat, wing is shown by the solid line on the left-hand side of the figure. For a plane wing the minimum drag occurs at zero lift and theoretically has a parabolic shape with the increment due to lift ΔC_D equal to a constant times the lift coefficient squared. In general, the data for the wings presented in this paper were fairly linear plotted against $C_L{}^2$ up to lift coefficients of about 0.3 and therefore the slope $\Delta C_D / C_L{}^2$ will be used to describe the drag-due-to-lift characteristics of plane wings in this lift range. For a cambered or cambered and twisted wing the drag curve, as shown by the dashed line, does not have its minimum at zero lift and therefore the drag polars will be used to describe the characteristics of this type of wing.

Now, if viscous forces are neglected, the drag due to the lift can be divided into two components - a thrust component of the suction force caused by the flow about the nose of the airfoil, and a drag component of the normal force. For a two-dimensional wing these two components exactly balance each other; however, for a three-dimensional wing the drag component of the normal force is greater than the thrust component of the suction force, since a higher angle of attack is required to develop the same lift, and an induced drag results. At subsonic speeds the rate of change of the induced drag with lift squared can be approximated by $1/\pi A$ as illustrated in the bottom part of the right-hand side of figure 1. Additional drag also occurs if the suction force is not fully developed at the leading edge. For the extreme case of zero suction the drag due to lift is equal to the component of the normal force, and the rate of change is therefore equal to the reciprocal of the lift-curve slope as illustrated in the top part of the figure. The drag curve of a wing usually lies somewhere between these two extremes and its relative position between these two limits is dependent to a large extent on the amount of suction developed at the leading edge and is therefore a function of such parameters as Reynolds number, Mach number, thickness, and leading-edge radius. The two limits, of course, are primarily a function of plan form and Mach number.

EFFECT OF REYNOLDS NUMBER

Figure 2 shows the effect of Reynolds number on the drag due to lift of an aspect-ratio-2 delta wing having an NACA 0005-63 airfoil section (ref. 3 and unpublished data). The results are presented in the form of the drag-rise parameter $\Delta C_D / C_L^2$ against Reynolds number for several Mach numbers. Also shown are the subsonic and the $M = 1.7$ theories for full leading-edge suction and the values for zero suction given by $1/C_{L_\alpha}$.

The results indicate that at a Mach number of 0.25 there is a rather large increase in drag due to lift with decreasing Reynolds number but that as the Mach number increases the effect of Reynolds number diminishes and is relatively unimportant at a Mach number of 1.7. The increase with decreasing Reynolds number is probably due in part to the fact that the combination of low Reynolds number and a relatively sharp leading edge is conducive to leading-edge separation resulting in a loss of leading-edge suction. In addition, a part of this variation is probably due to the fact that at low Reynolds numbers the transition point moves forward with increasing lift resulting in an increase in viscous forces with lift. The decreasing effect of Reynolds number with increasing Mach number is due to the fact that the difference between the theory and the zero-suction case decreases with increasing Mach number and the fact that the flow about the leading edge is affected by compressibility. It should be pointed out that, while the Reynolds number based on the mean aerodynamic chord was used here to define more clearly the variation with Reynolds number for a given wing, it appears that the drag due to lift at a given Mach number is more dependent upon the Reynolds number based on leading-edge radius. A recent correlation (ref. 4) based on this parameter succeeded in bringing the drag-due-to-lift parameter into fair agreement for a large number of aspect-ratio-2 delta wings having various airfoil sections. It should also be pointed out, however, that, for plan forms where compressibility effects are a function of thickness ratio or leading-edge radius, correlations based on the leading-edge Reynolds number would not be expected to bring the data into agreement at all Mach numbers.

EFFECT OF THICKNESS

Figure 3 illustrates the effect of wing thickness ratio on the drag-due-to-lift factor for unswept wings of aspect ratio 4 at Reynolds numbers of approximately 4×10^6 (refs. 5, 6, and unpublished data from the Langley 16-foot transonic tunnel). In addition to the experimental data, the theory for full suction is also shown. It will be noted that at subsonic speeds a decrease in thickness ratio from 8 percent to 4 percent increased the drag-due-to-lift factor; for example, at a Mach number of 0.6 it was increased by approximately 60 percent. This increase with decreasing thickness ratio is probably due to the fact that the 4-percent-thick airfoil section has a considerably smaller leading-edge radius and therefore develops less leading-edge suction. However, it will be noted that as the Mach number is increased the curves tend to converge and at a Mach number of about 0.88 there is little effect of thickness. This is due to the fact that, although the thick wing develops more suction at low speeds, the effect of compressibility on the flow about the leading edge is greater than for the thin wing. Above a Mach number of 0.88, the 4-percent-thick wing has considerably less drag due to lift than the 6-percent- and 8-percent-thick wings due to the fact that in this Mach number range the resultant force is normal to the wing chord, and since the thin wing has the higher lift-curve slope it has the lower drag due to lift. This is illustrated by the two dashed curves representing the reciprocal of the lift-curve slope for the 4-percent- and 6-percent-thick wings.

Figure 4 shows the effect of thickness on the drag due to lift of a delta wing of aspect ratio 2 at a Reynolds number of 3×10^6 (ref. 3). At subsonic Mach numbers it will be noted that the results are similar to those for the unswept wings (fig. 3) with the thin wing having the highest value of drag due to lift. However, at the higher Mach numbers the effect of thickness did not reverse for the delta wing as it did for the unswept wing and the thin wing still had the highest drag due to lift. It will also be noted that even at the highest Mach number tested the drag is lower than the reciprocal of the lift-curve slope, an indication of some suction being developed. This is due to the fact that the Mach number normal to the leading edge of this wing never exceeded a value of about 0.80. The vertical dashed line represents the free-stream Mach number for which the Mach number normal to the leading edge is equal to 0.9 which is approximately equal to the Mach number of the unswept wings for the case of zero suction. In order to indicate the variation with Mach number in the transonic range, the results of a rocket-propelled model of similar plan form having a thickness of $6\frac{1}{2}$ percent (ref. 7) is shown by the long and short dashed curve.

EFFECT OF LEADING-EDGE RADIUS

The effect of leading-edge radius on the drag due to lift of an unswept wing (ref. 3) is illustrated in figure 5. The wing had an aspect ratio of 3, a taper ratio of 0.39, and a thickness of 3 percent and was tested with a biconvex section and with a biconvex section modified with an elliptical nose having a radius of 0.045 percent of the chord. It will be noted that the results are similar to those obtained in the thickness investigation, with improvements with increasing leading-edge radius occurring only at subsonic speeds. It should be pointed out that the two curves are coincident at supersonic speeds.

Figure 6 presents the results obtained on a 45° swept wing of aspect ratio 4 which was tested with several modifications to the basic NACA 65A006 airfoil section in the Langley high-speed 7- by 10-foot tunnel. The three configurations tested were a sharp edge having zero radius, the normal radius of 0.24 percent chord, and a radius of 0.72 percent chord. Inasmuch as only a limited Mach number range was covered in these tests, the results are presented as ΔC_D plotted against C_L at a Mach number of 0.90. The results indicate that no improvement occurred with increase in the leading-edge radius at this Mach number.

EFFECT OF ASPECT RATIO

Figure 7 illustrates the effect of aspect ratio on the drag due to lift through the Mach number range. The wings were of delta plan form and 3 percent thick and had aspect ratios of 2 and 4 (ref. 3). The results indicate, as would be expected, that the higher aspect ratio has the lower drag due to lift throughout the Mach number range. However, it will be noted that the difference between the two aspect ratios is considerably greater than that indicated by the theory. It will be noted, however, that the effect of aspect ratio on the reciprocal of the lift-curve slope, which represents the zero-suction case, is approximately twice that for the full-suction theory at subsonic speeds. The larger effect of aspect ratio obtained in the experiments is therefore not surprising since these thin wings lose a good portion of suction.

EFFECT OF SURFACE SHAPE

The previous figures have illustrated the effect of various parameters on the drag due to lift of planar wings and have shown that, in general, the drag due to lift is considerably higher than the theoretical values largely because of separation at the nose and the accompanying loss of thrust. However, a theoretical study by Jones (ref. 8) has shown that an effective leading-edge thrust can be obtained by cambering and twisting the wing.

Figure 8 presents the results of cambering and twisting a 45° swept wing of aspect ratio 4 (ref. 9 and unpublished data from the Langley 8-foot transonic tunnel and the low-turbulence pressure tunnel). On the left-hand side of the figure the results are presented for the case of the cambered and twisted wing having 4.5° incidence at the fuselage which results in low fuselage angles at moderate lift coefficients. The results are presented as a plot of C_D against C_L at a Mach number of 0.9 for the plane wing at zero incidence, for the wing cambered and twisted for a uniform load distribution at a C_L of 0.4 and $M = 1.2$ ($\epsilon = 4.5^\circ$), and for the wing cambered and twisted for a triangular span load and a rectangular chord load at a C_L of 0.4 and $M = 0.9$ ($\epsilon = 13^\circ$). The triangular span load of the latter case was used in an attempt to improve the pitching-moment characteristics. The results indicate large increases in drag for both the camber and twist distributions. However, on the right-hand side of the figure, results are presented for the wing cambered and twisted for a uniform load tested on a slightly different fuselage but having approximately zero incidence at the fuselage. These results indicate substantial reductions in drag above a lift coefficient of about 0.15 for the cambered and twisted wing. For the case of zero incidence the fuselage is developing lift at the design condition and therefore the wing-fuselage combination represents the wing alone for which the camber and twist were designed considerably better than the configuration having 4.5° incidence which results in low fuselage angles in the moderate lift range.

Figure 9 presents the results of an aspect-ratio-2 delta wing tested with three different surface shapes (ref. 3): a planar surface, a surface cambered and twisted for a trapezoidal spanwise load distribution, and a surface which was planar over the inboard 80 percent of the local semispans. This third surface was a modification of the surface required for an elliptical span loading and was used in order to simplify construction. It should be mentioned that the wing incidence was zero at the plane of symmetry for all three cases. At a Mach number of 0.91, it will be noted that both types of surface modification resulted in improvements in the drag characteristics but that the simple nose camber was superior below a lift coefficient of about 0.3. As the Mach number increased the improvements diminished for both surfaces and at a Mach number of 1.53 no improvement was obtained; however, less penalty occurred for the simple nose camber.

Figure 10 shows the results obtained from tests in the Langley 8-foot transonic tunnel of a similar wing in which the extent of the nose camber was varied. The wing had an aspect ratio of 2.2 and a modified NACA 0004-65 airfoil section. Two nose cambers were tested and both were of constant chord, one being 4 percent (modification A) and the other 8 percent (modification B) of the mean aerodynamic chord. The camber covering 4 percent of the mean aerodynamic chord was obtained by shearing the ordinates so that the bottom surface was parallel to the chord line. The camber covering 8 percent of the mean aerodynamic chord was obtained by extending the chord and displacing the leading edge an amount equal to 1.3 percent of the longitudinal distance X from the wing apex. This was a modification of the surface shape required for an elliptical loading at a lift coefficient of 0.15 and was similar to that presented in figure 9 except that it was of constant chord. On the left-hand side of the figure drag polars are shown for the basic wing and the two modifications at a Mach number of 1.0. The results indicate that modification A (4 percent) had no effect on the drag while modification B (8 percent) resulted in a substantial reduction in drag except for an extremely small increase at zero lift. Although the coefficients are based on the actual areas, it should be pointed out that even the actual drag for a given lift is less for modification B than for the basic wing. The effect of Mach number on the variation of drag at a lift coefficient of 0.3 for the three configurations is shown on the right-hand side of the figure. It will be noted that both modifications resulted in improvements at the lower Mach number but that modification A had no effect above a Mach number of about 0.90; however, modification B resulted in improvements throughout the Mach number range investigated.

EFFECT OF TRIMMING

In order to reduce the weight and zero-lift drag of an aircraft, tailless configurations are sometimes used. However, since a tailless design, in general, obtains its trim from a surface on the wing, large deflections of this surface are required because of the short moment arm. These large deflections, of course, result in additional drag which could have an important effect on the performance. This is especially true at supersonic speeds because of the increased stability caused by the rearward movement of the wing aerodynamic center in going from subsonic to supersonic speeds and the higher drag due to flap deflection. Figure 11 illustrates this effect of trimming on the variation of the drag with lift. The model was an aspect-ratio-2 delta wing having an NACA 0005-63 airfoil and a constant-chord flap equal to 10 percent of the wing mean aerodynamic chord (ref. 10). At a Mach number of 0.90, it will be noted that a positive flap deflection of 4° resulted in a reduction in the drag due to lift. However, for a stable tailless configuration negative deflections are required to trim the airplane through the positive lift range which results in an increase in the drag due to lift. At a Mach number of 1.90, the increase due to trimming the airplane is considerably greater than at 0.90 because of the aforementioned increase in stability and drag due to flap deflection at supersonic speeds.

EFFECT OF APPLICATION OF THE AREA RULE

In references 1 and 2 it was shown that indentations of the fuselage according to the Mach number of 1.0 area rule resulted in large decreases in the zero-lift drag of wing-fuselage combinations at transonic speeds. The question now arises as to whether these benefits are maintained under lifting conditions. Figure 12 shows the effect of a Mach number of 1.0 body indentation on the drag of a wing-fuselage combination. The wing had an aspect ratio of 4, 45° of sweep, a taper ratio of 0.3, and an NACA 65A006 airfoil section and was tested in the Langley 8-foot transonic tunnel. The results are presented as total drag coefficient against Mach number for lift coefficients of 0 and 0.3 and indicate that the large reductions in drag at transonic speeds due to body indentation were to a large extent maintained in the lifting condition. At supersonic speeds the Mach number 1.0 indentation had negligible effect at either lift coefficients of 0 or 0.3.

Figure 13 shows the improvement in the maximum lift-to-drag ratio associated with this application of the area-rule concept. The results for both the basic configuration and the configuration with the indented body are plotted against Mach number and it will be noted that, below a Mach number of about 1.4, the lift-to-drag ratios were improved and at a Mach number of 1.0 (the design condition) the increase amounted to approximately 37 percent.

At the present time little has been done in attempting to develop area distributions which might actually reduce the drag increment due to lift. However, figure 14 presents the results of one such investigation conducted on a wing of aspect ratio 4 having 45° of sweep (ref. 11). The basic body was cylindrical rearward of the wing leading edge and was modified by several types of indentations. The first indentation, designated by the letter B in the figure, was symmetrical around the fuselage and was determined by the Mach number 1.0 area rule. The other two indentations tested were more abrupt indentations superimposed first on the upper half C and then on the lower half D of the symmetrical indentation. On the left-hand side of the figure the drag at zero lift is presented against Mach number and it will be noted that all the modifications gave about the same reduction in drag. On the right-hand part of the figure the same comparison is made for a lift coefficient of 0.3. The results indicate that the symmetrical area-rule indentation resulted in about the same reduction in drag as at zero lift which is consistent with figure 12. However, when the more abrupt indentations were added, additional reductions in drag resulted with the lowest occurring for the indentation below the wing.

COMBINED EFFECTS

At transonic speeds it has been shown that application of the area rule and the use of camber and twist results in significant reductions in drag. Figure 15 shows the effect of combining these two methods at transonic speeds. The tests were conducted in the Langley 8-foot transonic tunnel on a model having 45° of sweep, an aspect ratio of 4, a taper ratio of 0.6, and an NACA 65A006 airfoil section. The model was tested (1) with the basic wing and body, (2) with the basic wing and the body indented according to the area rule, and (3) with the wing cambered and twisted for a uniform load at $C_L = 0.4$ and $M = 1.2$ in combination with the indented body. On the left-hand side of the figure drag polars are presented for the three configurations at a Mach number of 1.0. It will be noted that indenting the fuselage resulted in large reductions in drag throughout the lift range. Camber and twist resulted in a rather large increase in minimum drag but resulted in improvements above a lift coefficient of about 0.2. On the right-hand side of the figure the maximum lift-to-drag ratios are plotted as a function of Mach number. At a Mach number of 0.8 the improvement is due mainly to the camber and twist and resulted in an increase from 13 to 17. At a Mach number of 1.0, the improvement is due mainly to the body indentation and resulted in an increase from about 7.5 to 11.5.

CONCLUDING REMARKS

In conclusion, it appears that Reynolds number has a rather large effect on the drag due to lift of thin wings at low speeds but that this effect decreases considerably with increasing Mach number.

Comparisons of wings of various thicknesses indicate that at subsonic speeds an increase in thickness is beneficial, whereas, in general, at transonic and supersonic speeds no gains and possible losses occur unless the wing leading edge is highly swept which results in relatively low subsonic speeds normal to the leading edge. Similar results are indicated with regard to leading-edge radius.

Although camber and twist are effective in reducing the drag due to lift at the design condition, providing the correct wing incidence is used, it appears that simple nose camber will result in similar gains with less penalty near zero lift.

The reductions in minimum drag associated with application of the area rule by means of fuselage indentations are maintained in the lifting condition and significant improvements in the lift-to-drag ratios result. In addition, from preliminary tests, it appears that local modifications to the fuselage indentations may result in additional reductions in drag at lifting conditions.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., September 3, 1953.

REFERENCES

1. Whitcomb, Richard T.: Recent Results Pertaining to the Application of the "Area Rule." NACA RM L53I15a, 1953.

2. Jones, Robert T.: Theory of Wing-Body Drag at Supersonic Speeds. NACA RM A53H18a, 1953.

3. Hall, Charles F.: Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. NACA RM A53A30, 1953.

4. Osborne, Robert S., and Kelly, Thomas C.: A Note on the Drag Due to Lift of Delta Wings at Mach Numbers Up to 2.0. NACA RM L53A16a, 1953.

5. Allen, Edwin C.: Experimental Investigation of the Effects of Plan-Form Taper on the Aerodynamic Characteristics of Symmetrical Unswept Wings of Varying Aspect Ratio. NACA RM A53C19, 1953.

6. Wiggins, James W., and Kuhn, Richard E.: Wind-Tunnel Investigation of the Aerodynamic Characteristics in Pitch of Wing-Fuselage Combinations at High-Subsonic Speeds. Sweep Series. NACA RM L52D18, 1952.

7. Mitcham, Grady L., Crabill, Norman L., and Stevens, Joseph E.: Flight Determination of the Drag and Longitudinal Stability and Control Characteristics of a Rocket-Powered Model of a 60° Delta-Wing Airplane From Mach Numbers of 0.75 to 1.70. NACA RM L51I04, 1951.

8. Jones, Robert T.: Estimated Lift-Drag Ratios at Supersonic Speeds. NACA TN 1350, 1947.

9. Harrison, Daniel E.: A Transonic Wind-Tunnel Investigation of the Characteristics of a Twisted and Cambered 45° Sweptback Wing-Fuselage Configuration. NACA RM L52K18, 1952.

10. Boyd, John W., and Pfyl, Frank A.: Experimental Investigation of Aerodynamically Balanced Trailing-Edge Control Surfaces on an Aspect Ratio 2 Triangular Wing at Subsonic and Supersonic Speeds. NACA RM A52L04, 1953.

11. Loving, Donald L.: A Transonic Wind-Tunnel Investigation of the Effect of Modifications to an Indented Body in Combination With a 45° Sweptback Wing. NACA RM L53F02, 1953.

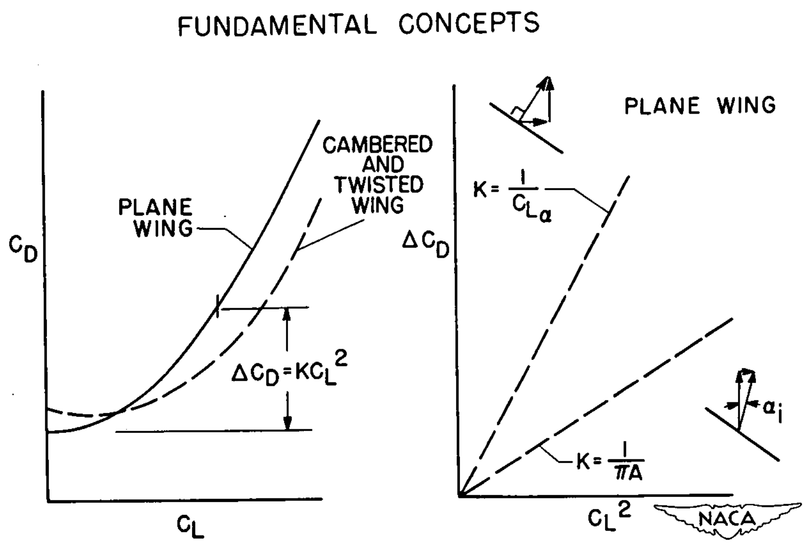

Figure 1

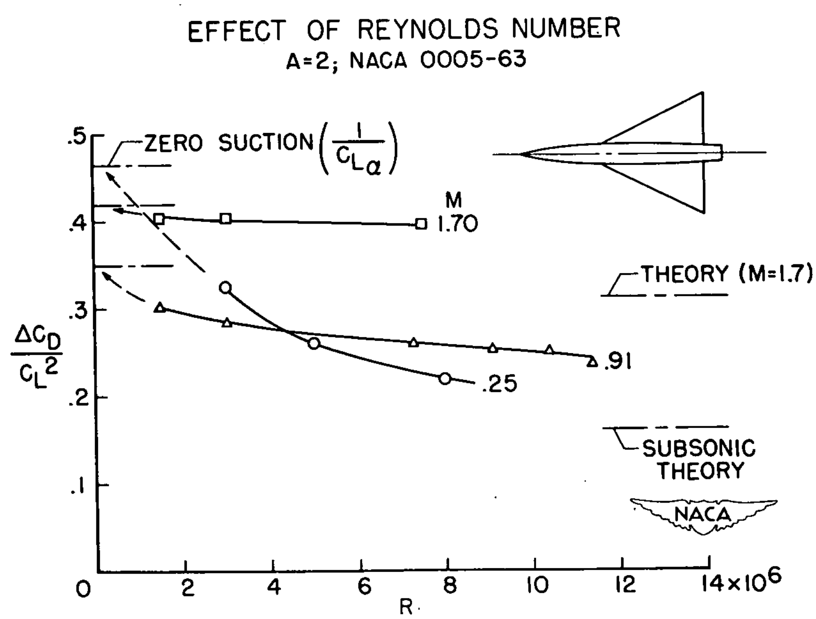

Figure 2

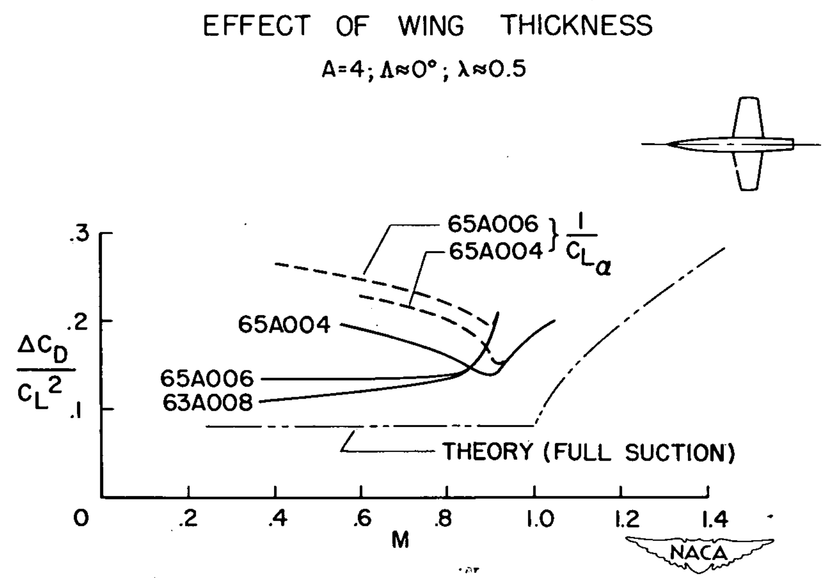

Figure 3

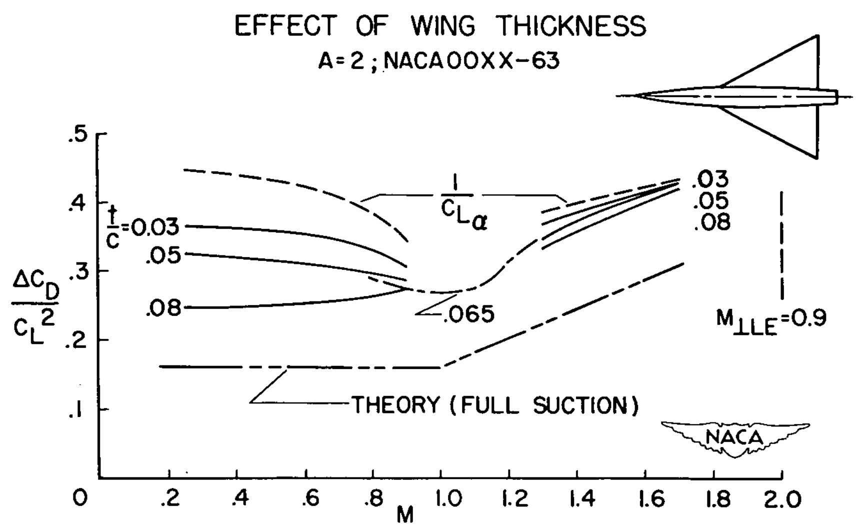

Figure 4

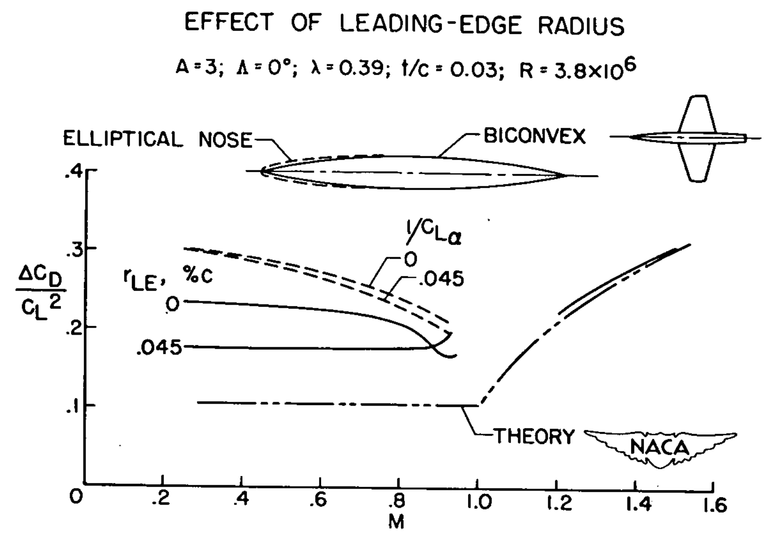

Figure 5

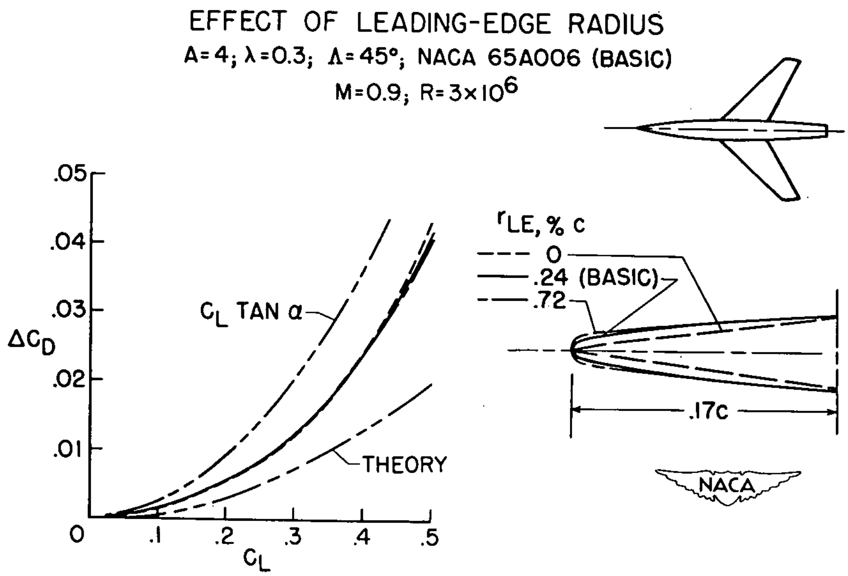

Figure 6

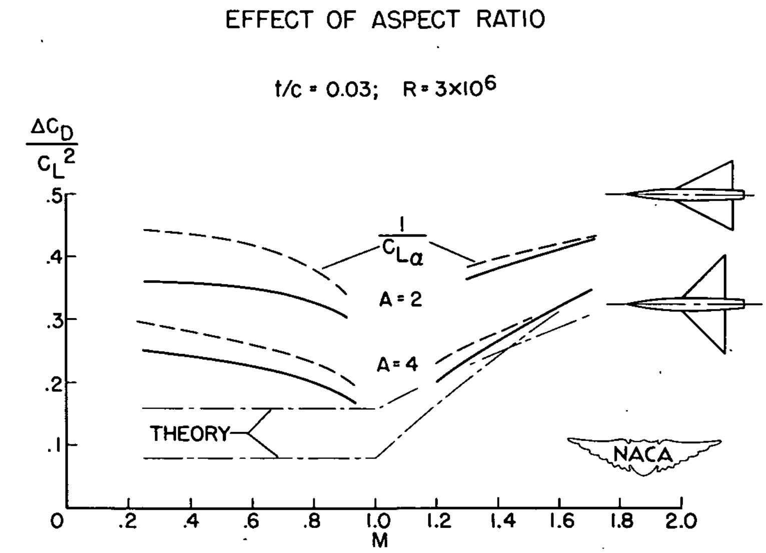

Figure 7

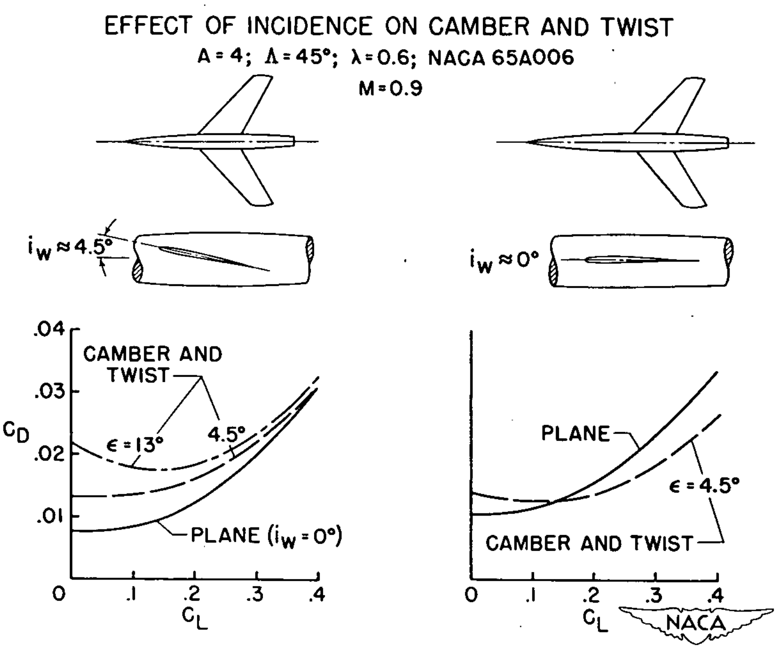

Figure 8

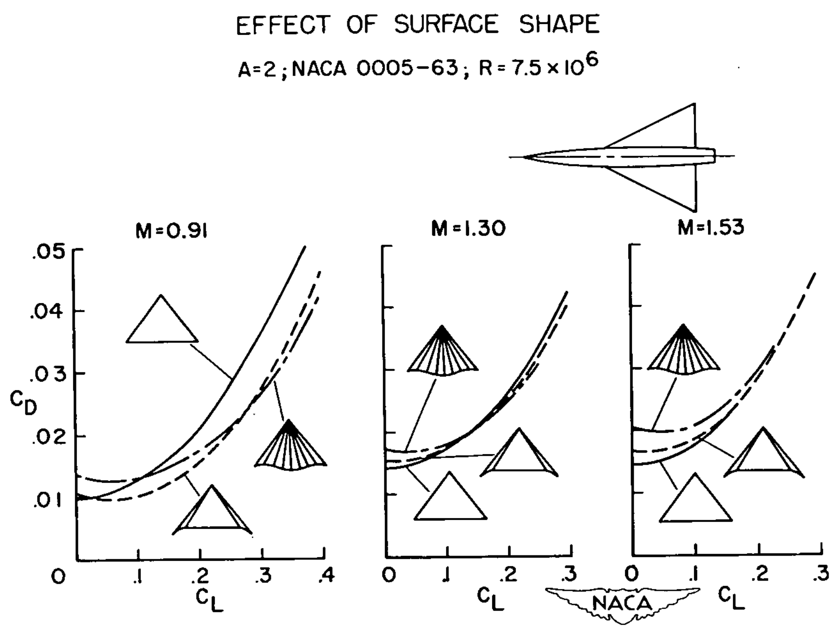

Figure 9

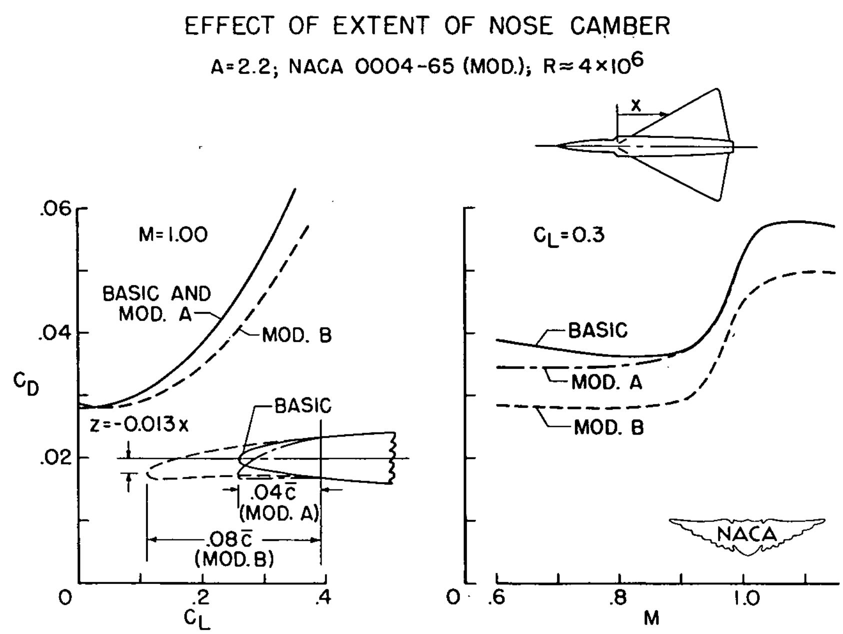

Figure 10

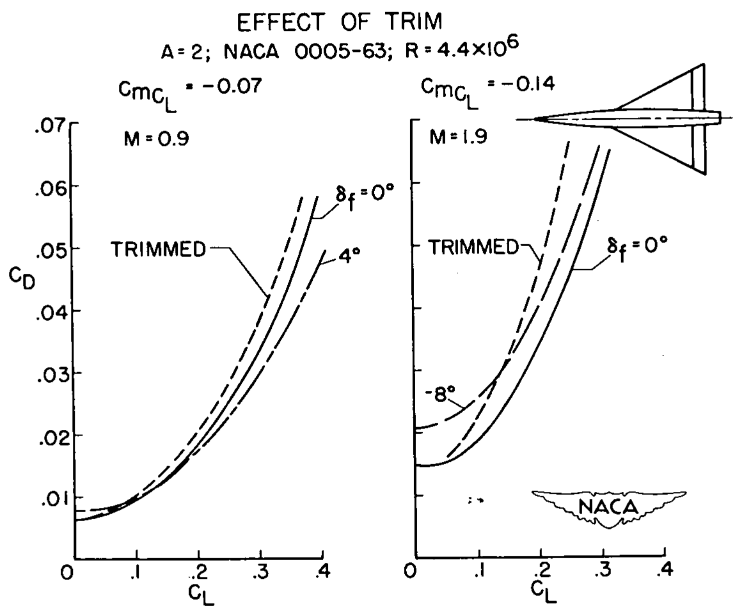

Figure 11

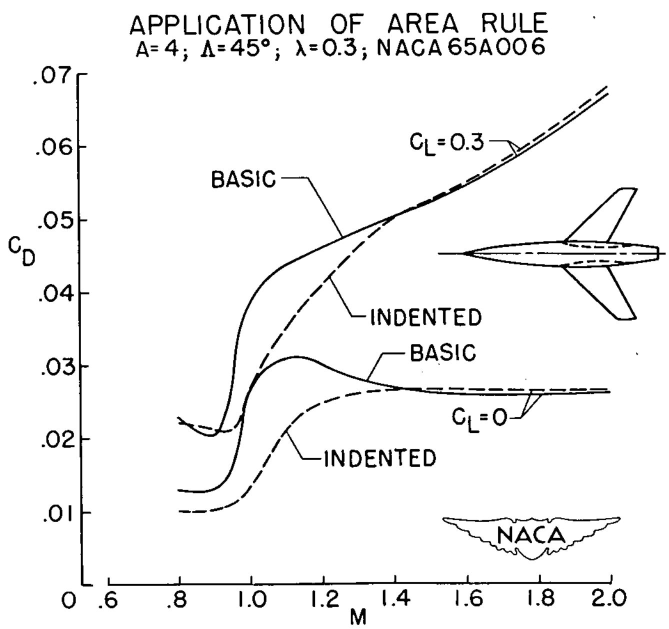

Figure 12

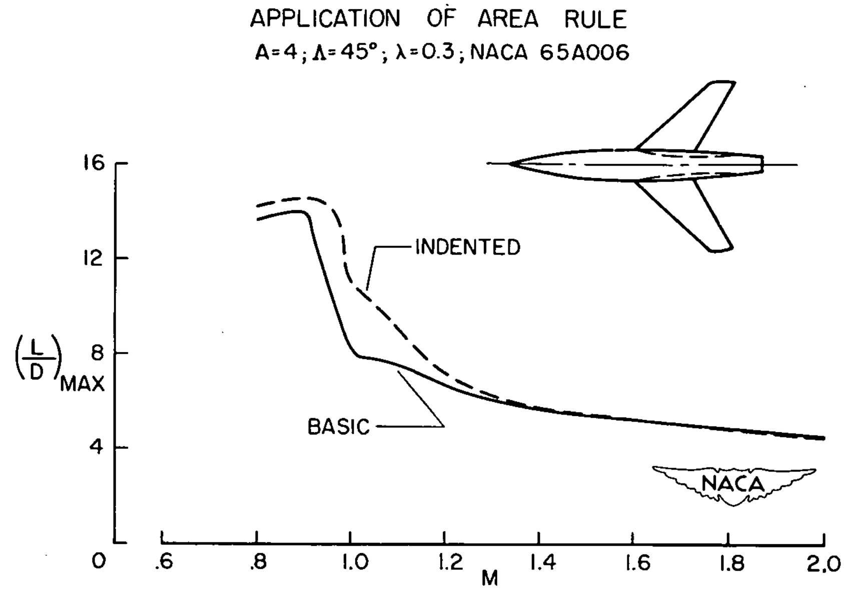

Figure 13

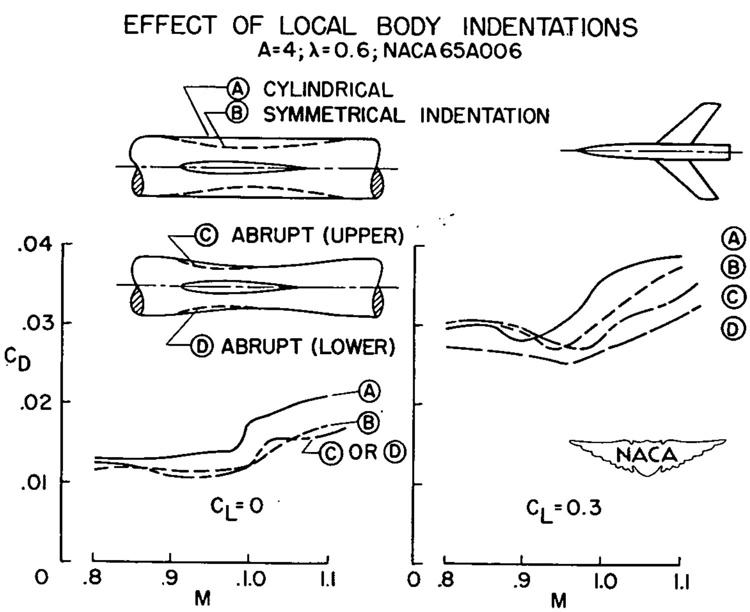

Figure 14

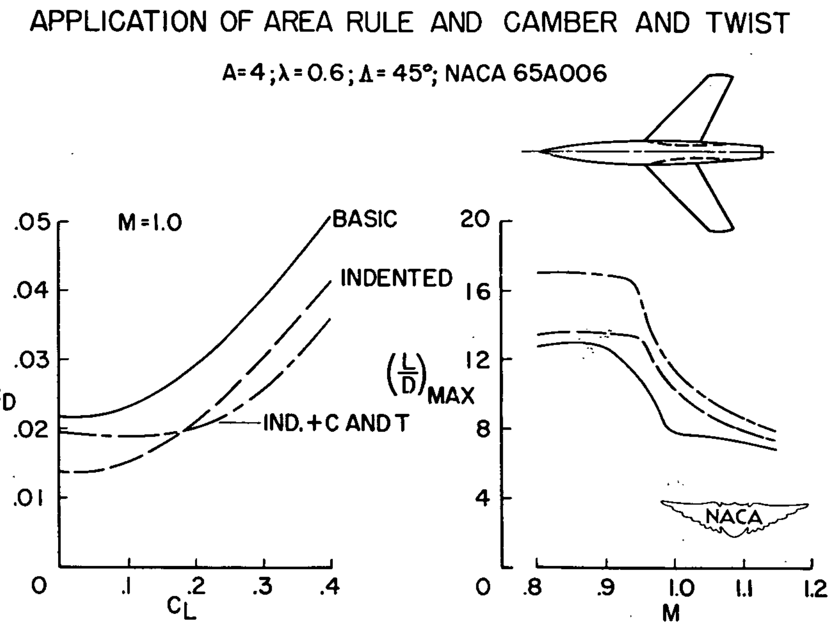

Figure 15

NACA-Langley - 11-18-53 - 325